Making of Wing Models by Tangent - Milling

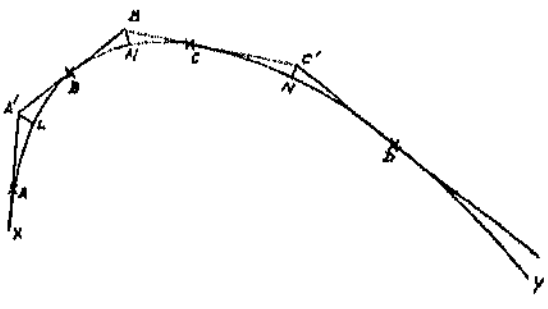
R. Sankar* and S, Janardhan*

ABSTRACT

Wings whose surfaces arc developable have been milled on a jig-borer at the National Aeronautical Laboratory, Bangalore, employing tangential milling. In this process, the Wing cross-section is approximated by a polygon which can be smoothened by hand-finish. The polygonal approximation itself is such that each side of the polygon is a tangent to the aerofoil. The aerofoil is defined by a finite sat of points got from experiments or otherwise. These point a are joined smoothly by using Spline approximation to achieve continuity of first and second derivatives. The splines and the settings of the jig-borer (for tangential milling) were obtained on the NAL SIRIUS Computer. Each setting of the jig-borer consists of the cutter-height and two turnings of the turnable one about the axis perpendicular to the turntable and the other about a fixed horizontal axis, so that the plane of milling becomes horizontal. The two angles of rotation and the cutter height depend upon the Wing geometry besides some of the. machine parameters.

THE PRINCIPLE OP TANGENT-MILLING

A convex developable surface lends itself very well to tangent-milling, To mill such a surface whose section is, say, XABCDY (Fig. 1), points such as A, B, C, D have to be located on the surface cross-section and milling done in the direction of the tangent planes at these points. If the intersections of the tangent planes arc along straight line generators through A′, B′, C′, the surface cross section ABCD could be considered approximated by the polygon AA' BB' CC' D. The closeness of this approximation depends on the smallness of the ridge heights A′ L, B′M, C'N, where L,M,N are the feet of the normals from A′, B′, C′.



Fig,1

To reproduce the convex surface from the above approximation ridge heights like A′L, B'M, C'N have to be removed by hand-finish,

THE PLANES OP MILLING

The fixation of the tangential planes of milling (such as AA′, A'BB', B′CC, C′D) is rendered difficult if the equation of the surface cross-section is either not known or not simple. This is partly because of its dependence on the prescribed ridge heights A'L, B'M, C'N. The equations resulting from imposition of such conditions are not easy to handle.

In the case of aeroplane wings, the aerofoil section is usually given as a finite set of points, its equation being unknown. Since analytical considerations are ruled out, the location of the tangential planes of milling from a knowledge of the coordinates of such points is possible only through appropriate numerial techniques, using a digital computer. This is described below.

A COMPUTATIONAL PROCEDURE POR FIXING THE TANGENTIAL PLANES OF MILLING POR A WING SECTION

P_1, P_2, P_3, P_4 etc. arc given points on an aerofoil section contour (Fig. 2), additional points between these can be generated by fitting a Spline through them. A spline replaces the actual curve by piecewise polynomials. Every pair of given points will he joined by a polynomial and at the junction points derivatives upto a certain order will be matched. The resulting spline will thus be a smooth curve, the degree of smoothness depending upon the order of the highest derivation whose continuity is desired.

*National Aeronautical Laboratory, Bangalore.

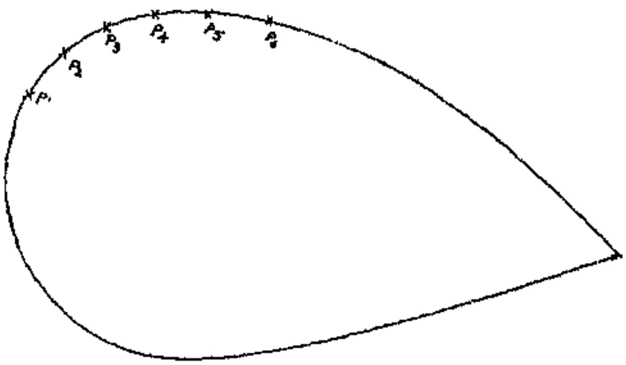


Fig. 2

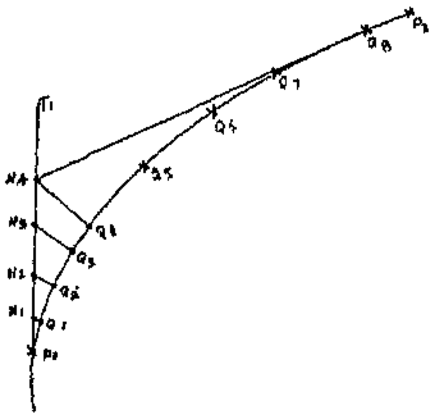


Fig. 3

Let $Q_1, Q_2 \ldots$ be additional points between say P_1 and P_2 (Fig. 3). The coordinates of these points and the slopes of the tangents thereat can be got from the spline approximation.

Let $P_1 T_1$ be a tangent to the aerofoil at P_1. If the tangential plane to the wing through $P_1 T_1$ is selected as one plane of milling, the next plane of milling can be fixed as follows : At $Q_1, Q_2, Q_3 \ldots$ the normal lengths $Q_1 N_1, Q_2 N_2, Q_3 N_3$ intercepted between the section contour and tangent. $P_1 T_1$ are compared with the permitted ridge height S, which is a priori given. One of the points say Q_4 has a normal length Q_4 N just less than S while at the next point Q_5 the normal exceeds S. The tangent to the curve passing through N would then fix the next plane of milling. To fix the position of this tangent, we have to scan the points beyond Q_4, one by one, to check whether the point N lies to the right or left of the tangents at $Q_5, Q_6 \ldots$ Initially, N would lie to the left of the tangent at say Q_5, the positive direction of the tangent being taken in the conventional sense. As we pass from a point like Q_7 to a point like Q_8 the point N would cross over from the left to the right of the tangent at these points. We then accept the tangent Q_7 N as fixing the next plane of milling. In similar manner the other tangential planes of milling can be fixed.

EMPLOYMENT OF TANGENT-MILLING ON A MILLING MACHINE

In a milling machine, such as the jig borer at N.A L., any job that has to be milled to a desired shape is mounted on a turntable. The turntable can be rotated about an axis through its centre perpendicular to itself. Rotation about another horizontal axis fixed relative to the machine is also provided.

To obtain a wing shape by tangent milling, starting from a metal blank, the blank is mounted on the turntable. Since the mill-cutter can move only in a horizontal plane, each tangential plane of milling described earlier has to be rotated to the horizontal position. This is achieved by rotating the job about the two axes of rotation in turn. The cutter is then moved to the position, corresponding to the now-horizontal tangent plane of milling.

The two angles of rotation of the turntable and the cutter-height, corresponding to each tangential plane of milling, depend upon the geometry of the wing and some constant parameters of the milling machine. In the NAL jig-borer, rotation about the axis perpendicular to the turntable is through the full range of 360°, while that about the fixed horizontal axis is through a range of 90° only. The latter restriction may necessitate mounting the job on wedge placed on the turn-table, if the geometry of the wing warrants this

AN ACTUAL EXAMPLE

In what follows the results of applying the procedures outlined above to a wing depicted in Fig. 4 are presented :

The wing-shape, to be obtained from a metal blank is ABCDEFA. All sections of the wing between the chords AB and EG conform to the symmetric NACA 65A006 aerofoil. Between EG and DC, the aerofoils are modified versions of NACA 65A006 with a droop at the leading edge. The nose droop which introduces local asymmetry is such that the

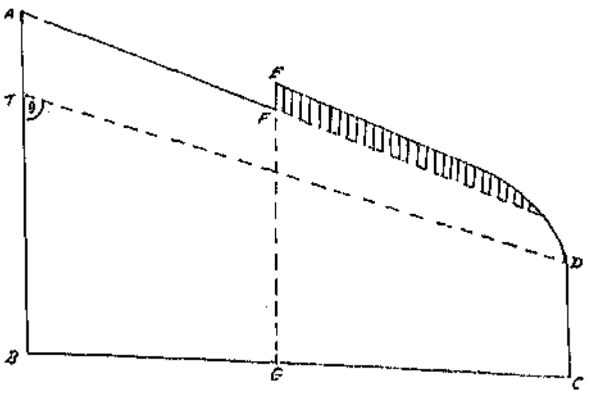


Fig. 4

new chord length is 10% more than the original chord length, the shape at the nose being similar to that of the unmodified NACA 65A006 aerofoil. The leftmost vertical tangent at the droop makes an angle of 4° with the tangent at the nose of the droop.

In Fig. 4 the droop portion of the wing is shown by the striped region. Chords AB and DC and the distances of DC and EG from AB are known. The chords vary linearly in the portions on either side of EFG, except near the edge DC. Likewise the thickness variation is also linear except near DC. The line TT' joining all the 25% points of the unmodified chords makes a prescribed angle 0 with AB.

Figures 5 and 6 give an idea of how the aerofoils on either side of EFG look like. The coordinates of a set of points on the aerofoil contour of Fig. 5 are given in Table 1. At the nose of this, aerofoil, the tangent is vertical, the contour near the trailing edge is almost linear.

The shape of the nose-droop of the modified aerofoil of Fig. 6 is not known, except at the nose, where the radius is assumed as 0.81 times that of the nose of the unmodified aerofoil. Actually the droop can be considered as obtained by moving the nose of the unmodified aerofoil (fig. 5) forward by 10% of the chord length and then tilting it down by 4°, so that the droop merges smoothly with the rest of the unmodified aerofoil. Mathematically the droop can be replaced by splines which satisfy the conditions prescribed at the nose and the points of merging.

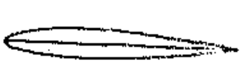


Fig. 5

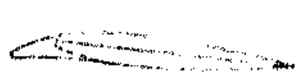


Fig. 6

SPLINE-APPROXIMATIONS TO THE SYMMETRIC AND MODIFIED NACA 65A006 AEROFOILS

The mathematical details of these spline-fits are described in another paper by the authors (Reference 1). The results are shown in Tables 2, 3 and 4 for the case where the wing of figure 4 has the following dimensions :

AB = 277.045 mm.

CD=1AB

Distance of DC from AB = 274.324 mm.

Distance of EFG from AB = 178.31 mm.

Angle e = 40°.

Table 2 corresponds to the symmetric NACA 65A006 aerofoil's upper contour. Tables 3 and 4 correspond to the top and bottom contours of the modified aerofoil. Table 4 indicates the presence of a concavity for the bottom. The tables clearly show that the spline approximations ensure continuity of the ordinate, first derivative and second derivative.

THE TANGENT-PLANES OF MILLING FOR THE SYMMETRIC AND MODIFIED NACA 65A006 AEROFOILS

The spline-approximations obtained in the preceding section were used to generate additional points on the aerofoil contours. The procedure already described (to fix the tangential planes of milling) was used to obtain a set of points on the aerofoil contours at which milling is to be clone tengentially. The coordinates of such points arc given in Tables 5, 6 and 7, these being obtained from Tables 2, 3 and 4 respectively.

Since a concave shape cannot be obtained by tangent-milling only the convex parts on either side of the concavity were milled for the bottom contour of the modified aerofoil. Care was also taken to ensure that the milling of one convex part did not cut into the other.

MACHINE-SETTINGS

As already stated, each point of Tables 5, 6 and 7 is realized on the milling machine in terms of two angles of rotation and the cutter height. These three quantities can be obtained as follows :

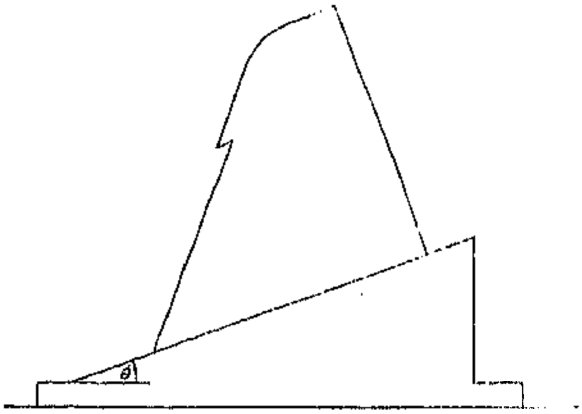


Fig. 7

Let the wing be mounted on a wedge of angle ϕ placed on the turntable (Fig. 7). If α denotes the angle of rotation of the turntable about its axis and p the angle of rotation about the fixed horizontal axis, then it can be shown from geometrical conside-rations that

$$\tan \alpha = g \cos \phi + A \sin \phi$$

$$\cos p = \frac{A \cos \phi - g \sin \phi}{\sqrt{1 + g^2 + \lambda^2}}$$

Height above horizontal axis of cutter

$$[c + (k + x_1) \sin \phi] \cos p + (N - x_1 \cos \phi) \sin a + y_1 \cos a] \sin p$$

where,

(i) (x_1, y_1) are the coordinates of the point on the aerofoil, selected for tangent-milling; (and referred to the nose as origin)

(ii) g is the slope of the aerofoil at (x_1, y_1)

(iii) $\lambda = \frac{3}{4} \frac{y_1}{M} + g \left[\frac{3}{16} \frac{L}{M} + \cot\theta - \frac{3}{4}\frac{x_1}{M}\right]$

(iv) 0 is the angle shown in fig. 4,

(v) L is the length of AB in fig. 4

(vi) M is the distance between AB and DC in fig. 4.

(vii) K Sin ϕ, the height of the leading edge of the root-section (taken as the origin) above the base of the wedge.

(viii) c, the height of the wedge base above the fixed horizontal axis.

(ix) N, the horizontal distance of the origin from the initially vertical axis of the turntable.

The machine-settings corresponding to Tables 5, 6 and 7 were computed for specific configurations and were used in the actual job

ACKNOWLEDGEMENT

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REFERENCE

K. SANKAR and S JANARDHAN : "Spline-approximation to an aerofoil section." NAL Technical Memorandum.

TABLE 1

Coordinates* of given data-points for the NACA 65A006 symmetric 6% Thickness airfoil

X	Y
0	0
0.5	0.464
0.75	0.563
1.25	0.718
2.5	0.981
5.0	1.313
7.5	1.591
10.0	1.824
15.0	2.194
20 0	2.474
25.0	2.687
30.0	2 842
35.0	2 945
40.0	2.9%
45.0	2.992
50.0	2.925
55.0	2.793
60.0	2.602
65.0	2.364
70 0	2.687
75.0	1.775
80.0	1.437
85.0	1.083
90.0	0 727
95.0	0.370
100.0	0.013

*Origin taken at the nose. Chord length assumed to be 100 units. Leading edge radius is 0.229% of chord.

TABLE 2

Spline-approximation to **the** upper **half** of **the** NACA 65A006 **symmetric** airfoil of Table 1

X	Y	dy/dx	d^2y/dx^2
0.00022	0.01000	22.89828	—52441.60
0.00087	0.02000	11.44656	.....6555 1220
0.00197	0.03000	7.62818	.....1942.2528
0.00350	0.04000	5.71813	.....819.38190
0.00546	0.05000	4.57144	-- 419.51708
0 02192	0.10000	2.27318	52.41987
0.04953	0.15000	1.50217	15.50798
0,08859	0.20000	1.11366	6.51772
0,13946	0.25000	0,87880	· · 3,31339
0.20252	0.30000	0.72138	• 1.89588
0.27822	0.35000	0.60872	.......1.17494
0.36693	0,40000	0.52445	· • 0.77094
0.46899	0.45000	0.45942	· • 0.52803
0.49999	0.46400	0.44384	--• 0.47797
0.74999	0.56300	0.35844	- 0.25034
1.05566	0.66300	0.29892	· · 0.17391
1,25000	0.71800	0.26838	.......0.13518
1.65660	0.81800	0.22622	· · 0,08997
2.14249	0.91800	0.18715	· - 0.06839
2.50001	0.98100	0.16659	· 0.04643
3.15720	1.08100	0.13974	- 0.03004
3.91877	1,18100	0.12543	· • 0.00992
4.99989	1.31300	0.11936	• 0.00698
5.85550	1.41300	0.11459	.....0 00603
7.49971	1.59100	0.10148	· 0,00720
8,50000	1.68901	0.09459	· · 0.00656
9.00000	1.73550	0.09142	· 0.00611
9.50000	1,78047	0.08847	· 0.00572
10.00000	1.82400	0.08568	·· 0 00550
11.00000	1.90705	0.08051	··· 0.00488
12.00000	1.98520	0.07585	-- 0.00447
13.00000	2,05887	0.07154	--- 0.00417
14,00000	2.12837	0.06751	---• 0.00388
15.00000	2.19400	0,06380	- 0,00351
16.00000	2.25611	0.06047	- 0.00318
17.00000	2.31503	0.05739	— 0,00302
18.00000	2.37092	0.05441	- 0,00294
19.00000	2.42387	0,05152	— 0.00284
20 00000	2.47400	0.04877	— 0.00263
21.00000	2.52149	0.04624	— 0.00245
23.00000	2 60911	0.04138	— 0 00244
25.00000	2.68700	0 03655	........0.00231
27,00000	2,75553	0.03202	— 0.00220
29.00000	2.81527	0.02776	— 0.00207
31.00000	2,86667	0.02366	.....0.00202
33.00000	2.90994	0.01959	— 0,00206
35 00000	2.94499	0.01545	— 0.00207
37.00000	2.97172	0.01126	— 0 00210
39.00000	2.99003	0.00704	• • 0,00213
41.00000	2,99968	0.00272	.......0.00217
43.00000	3,00060	— 0.00186	— 0.00244
45 00000	2.99193	........0.00693	— 0,00254
47.00000	2.97302	—0.01206	.....0.00260
49.00000	2.94365	—0.01732	— 0.00265
52.00000	2.87.980	—0.02519	....- 0.00257
55,00000	2.79298	-- 0.03260	--- 0.00138
58.00000	2.68453	-- 0.03942	- 0.00203
61.00000	2.55785	• -0.04506	· · 0.00179
64.00000	2 41492	- 0 05013	0.00160
67 00000	2,25820	· 0.05463	· 0.00146
70.00000	2.08760	· 0.05910	·· 0.00142
72.00000	1.96620	· 0.06188	• • 0.00130
74. 00000	1,83970	-0.06427	• · 0.00110
76.00000	1.71020	-- 0.06632	--- 0.00095
78.00000	1,57550	---0.06809	... 0.00083
80 00000	1.43770	·.....0 06960	· 0.00065
82.00000	1.29730	---0.07057	· · 0 00037
84.00000	1,15500	-- · 0,07114	· 0 00019
86,00000	1.01180	.....0.07126	- · 0,00000
88,00000	0 87000	——0.07125	--- 0 00002
90.00000	0,72760	· 0.07133	· 0 00004
95.00000	0.37000	· 0 07133	--- 0.00004
100.00000	0.01300	-- 0.07133	— 0 00004

TABLE 3

Spline-approximation to the upper contour of **the** modified 5.5% thickness NACA **65A006** aerofoil (X and Y **referred** to nose of the unmodified aerofoil)

X	Y	dy/dx	d^2y/dx^2
-9.9996	-1.7481	-14.3007	18901.735
-9.9835	-1.6643	2.2389	-66.4166
-9.9588	-1.6212	1.4334	-16.9157
-9.8749	-1.5327	0.8374	-3.2188
-9.7422	-1.4407	0.5929	-1.0870
-9.5577	-1.3451	0.4612	-0.4757
-9.2839	-1.2325	0.3734	-0.2215
-8.7940	-1.0700	0.2959	-0.1177
-7.7940	-0.8203	0.2161	-0.0419
-7.3500	-0.7273	0.2049	-0.0083
-6.3500	-0.5265	0.1967	-0,0082
-4.8500	-0.2405	0.1850	-0.0080
-2.8500	0.1128	0.1688	-0.0078
-0.8500	0.4348	0.1533	-0.0076
1.6500	0.7945	0.1345	-0.0074
4.1500	1.1080	0.1164	-0.0072
7.3000	1.4396	0.0943	-0.0068
7.5000	1.4584	0.0930	-0.0066
10.0000	1.6720	0.0785	-0.0050
13.0000	1.8873	0.0656	-0.0038
16.0000	2.0681	0.0554	-0.0029
19.0000	2.2219	0.0472	-0.0026
23.0000	2.3917	0.0379	-0.0022
27.0040	2.5259	0.0294	-0.0020
31.0000	2.6278	0.0217	-0.0019
35.0000	2.6996	0.0142	-0.0019
40.0000	2.7462	0.0045	-0.0020
45.0000	2.7426	-0.0064	-0.0023
50.0000	2.6812	-0.0183	-0.0024
55.0000	2.5602	-0.0299	-0.0022
60.0009	2.3851	-0,0396	-0.0017
65.0000	2.1672	-0.0474	-0.0014
70.0000	1.9134	-0.0542	-0.0013
75.0000	1.6274	-0.0599	-0.0010
80.0000	1.3180	-0.0638	-0.0006
85.0000	0.9935	-0.0653	0
90.0000	0.6670	-0.0654	0

TABLE 4

(X and Y referred to nose of the unmodified aerofoil)

X	Y	dy/dx	d^2y/dx^2
-9.9996	-1.7481	-14.3007	18901.735
-9.9938	-1.7808	-3.4947	289.523
-9,9720	-1.8289	-1.5961	30.1009
-9.9416	-1.8681	-1.0761	10.0440
-9.8461	-1.9442	-0.6235	2.3740
-9.7019	-2.0168	-0.4176	0.8804
-9.5059	-2.0858	-0.3011	0.4057
-9.2191	-2.1592	-0 2212	0.1957
-8.7191	-2.2476	-0.1371	0.1409
-8.2191	-2,3008	-0.0804	0.0861
-7.7191	-2.3325	-0.0510	0.0312
-7.5000	-2.3430	-0.0469	0.0073
-6.5000	-2.3863	-0.0399	0.0068
-5.5000	-2.4228	-0.0333	0.0064
-4.0000	-2.4658	-0.0242	0.0057
-2.5000	-2.4958	-0.0160	0.0051
-0.5000	-2.5182	-0.0067	0.0042
1.5000	-2.5237	0.0009	0.0034
3.5000	-2.5157	0.0068	0.0025
6.0000	-2.4921	0.0117	0.0014
8.5000	-2.4596	0.0139	0.0(.03
11.5000	-2.4183	0.0130	-0.0010
14.5000	-2.3855	0.0082	-0 (023
17.5000	-2.3731	-0.0005	-0.0035
20.0000	-2.3848	-0.0104	-0.0046
23 000	-2.4311	-0.0209	-0.0027
26.0 0	-2.5028	-0.0259	-0.0007
30.000	-2.6052	-0.0236	0.0019
31.000	-2.6278	-0.0217	0 0019
35.000	-2.6996	-0.0142	0.0019
40.000	-2,7462	-0.0045	0.0020
45.000	-2.7426	0.0064	0,0023
50.000	-2.6812	0.0183	0.0024
55.000	-25602	0.0299	0.0022
60.000	-2.3851	0.0396	0.0017
65.000	-2.1672	0.0474	00014
70.000	-1.9134	0.0542	00013
75.000	-1.6274	0,0599	0.0010
80.000	-1.3180	0.0638	0.0006
85.000	-0.9935	0.0653	0
90.000	-0.6670	0.0654	0

TABLE 5

Points on the aerofoil of Table 2 selected for tangent-milling

X	Y	Slope
0	0	Infinite
0.0002	0.0100	22.89828
0.0219	0.1000	2.27318
0.0886	0,2000	1.11366
0,2025	0.3000	0.72138
03669	0.4000	0.52445
0.7500	0.56300	0.35844
1.2500	0.7180	0.26838
2.1425	0.9180	0.18715
3.1572	1.0810	0.13974
4.9999	1.3130	0 11936
8.3000	1.6699	0.09592
11.8000	1.9699	0.07676
16.0000	2.2561	0.06047
21.0000	2.5215	0.04624
26.0000	2.7224	0.03426
32.0000	2.8893	0.02164
37.5000	2.9771	0.01021
43.0000	3,0006	-0.00186
47.5000	2.9667	0.01337
52.5000	2.8669	-0 02646
57.5000	2.7041	0.03839
63.5000	2.4398	0 04932
70.5000	2.0573	0 05983
78.0000	1.5755	0.06809

TABLE 6

Points or. the Upper contour of the aerofoil of Table 3, selected for tangent-milling

(X, Y referred to nose of [he unmodified aerofoil)

X	Y	Slope
-9,9996	-1.7481	- 14,3007
-9.9971	-1.7066	5.21729
-9.9588	1.6212	1.43344
-9.8749	1.5327	0.83739
-9.7422	1.4407	0,52294
-9.5577	1.3451	0.46117
-9.2839	1.2,325	0.37336
-8.7940	1.0609	0,29592
- 7.7940	-0.8203	0.21610
-5.3500	-0.3338	0.18861
-2.3500	0.1962	0.16486
0.6500	0.6562	0.14198
3.6500	1.0489	0 11996
6.6500	1.3768	0.09880
10.0000	1.6720	0.07854
14.2000	1.9633	0.06118
21.0000	2.5215	0 04624
26.0000	2.7224	0.03426
32.0000	2.8893	0.02164
37.5000	2.9771	0.01021
43.5000	3 0006	-0.00186
47.5000	2.9666	-0.01337
52.5000	2,8669	-0 02646
57.5000	2.7041	-0.03839
63.5000	2.4398	-0.04932
71.5000	2.0573	-0.05983
78.0000	1,5755	-0.06809

TABLE 7

Point on the lower contour of the **aerofoil** of Table 4 **selected for tangent-milling** (X, Y refer to **nose** of the **unmodified** aerofoil)

X	Y	Slope
- 9.9996	-1.7481	-14.3007
- 9.9938	1.7808	- 3.49471
9,9416	1.8681	- 1 07611
9.8461	1.9442	- 0,62348
9.7019	2.0168	0.41760
9.5059	2.0838	0.30112
9.2191	2.1592	0.22121
8.7191	2.2476	0.13709
8.2191	2.3008	0.08036
6.5000	2.3863	0.03986
3,5000	2.4771	0.02134
0.5000	2.5182	0.00670
39.5000	2.7438	0.00547
43.0000	3,0006	0 00186
47.50000	-2 9666	0 01337
52.50000	- 2.8669	0.02646
- -57.50000	- 2.7041	0.03839
63.50000	2.4398	0.04932
-70,50000	20573	0,05983
78.0000	- 1.5755	0,06809